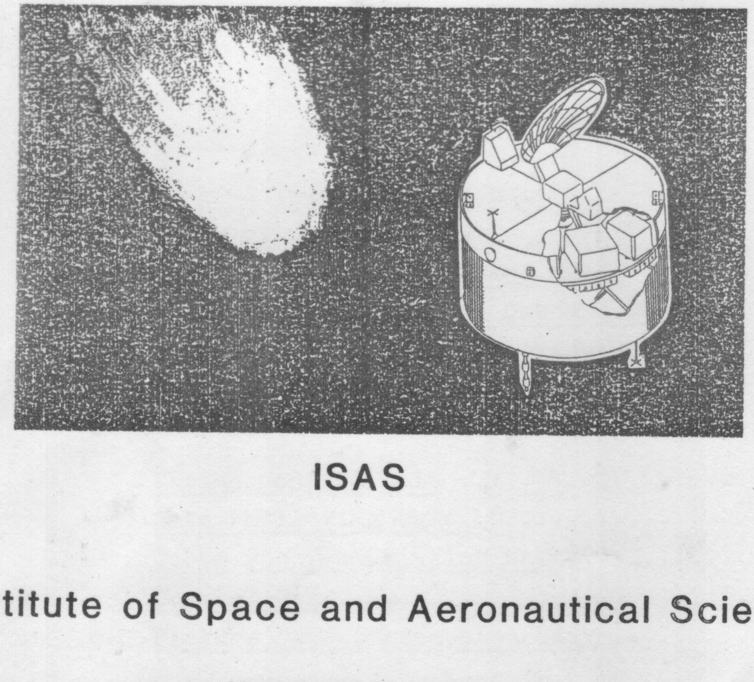
PLANET-A

JAPANESE FIRST INTERPLANETARY FLIGHT

STATUS REPORT



ISAS

Institute of Space and Aeronautical Science

November 1980

CONTENTS

Page

INTRODUCTION 1

SCIENTIFIC OBJECTIVES 2

ORBIT DESIGN 4

- HELIOCENTRIC TRAJECTORY 5
  - Planet-A 5
  - MS-T5 5
- GEOCENTRIC TRAJECTORY 7

LAUNCH VEHICLE 8

SPACECRAFT 9

- DESIGN PHILOSOPHY 9
- WEIGHT AND CONFIGURATION 11
- SCIENTIFIC INSTRUMENTS 14
  - Lyman-α Imaging for Halley's Comet 14
  - Solar Wind Measurement Onboard Planet-A 15
  - Plasma Wave Probe Onboard MS-T5 17
  - Solar Wind Measurement Onboard MS-T5 18
  - Measurement of the Interplanetary Magnetic Field Onboard MS-T5 19
- DATA PROCESSOR 20
  - Timing Control Unit 20
  - Data Formatting Unit 20
  - Coded PCM Transmission Unit 22
  - Control Unit 22
  - Picture Data Compression Unit 22
- COMMUNICATION SYSTEM 22
- RF SUBSYSTEM 24
- ANTENNA SUBSYSTEM 31
- HOUSE-KEEPING SUBSYSTEM 33

Page

ATTITUDE AND VELOCITY CONTROL 34

Sensors and Wheel 34

Sun sensor 34

Star scanner 36

Momentum wheel 37

Gas Jet Subsystem 37

TIMER AND IGNITION SYSTEM 40

POWER SUBSYSTEM 44

STRUCTURE 46

THERMAL CONTROL 47

Design Concept 47

Design Configuration 47

ENVIRONMENTAL REQUIREMENTS 49

MECHANICAL ENVIRONMENT 49

Test 49

MAGNETIC ENVIRONMENT OF MS-T5 51

SCHEDULE OF DEVELOPMENT 51

ILLUSTRATIONS

Figure Page

1 Heliocentric Orbit of PLANET-A and MS-T5 ... 6

2 Range between the Earth and the Spacecraft and the Sun-Earth-Spacecraft Angle (PLANET-A) .. 7

3 Cutaway View of Mu-3U 9

4 PLANET-A Spacecraft 12

5 PLANET-A System Block Diagram 13

6 UV Camera Block Diagram 16

7 Plasma Wave Probe Block Diagram 18

8 Data Processor Block Diagram 21

9 S/C → DSN Telemetry Link (Down Link) 27

10 S/C → KSC Telemetry Link (Down Link) 28

Figure Page
11 S/C ← DSN Command Link (Up Link) 29
12 S/C ← KSC Command Link (Up Link) 30
13 HK Subsystem Block Diagram 33
14 RCS Block Diagram 38
15 RCS Configuration 39
16 RCS Engine Arrangement and Operation 39
17 Timer and Ignition System Block Diagram 42
18 Interface of the Timer-Ignition System 43
19 Power Subsystem Block Diagram 45
20 Structure 46
21 Thermal Subsystem and Coatings 48

TABLES

Table Page
1 Nominal Orbital Parameters of PLANET-A and MS-T5 5
2 Injection Parameters of PLANET-A and MS-T5 8
3 Mu-3U Specification 10
4 PLANET-A Weight Breakdown 14
5 Parameters of Data Processor 23
6 Communication System Parameters 25
7 Event Sequence 35
8 Performance of RCS 40
9 Power Subsystem Performance 45
10 Sinusoidal Vibration 50
11 Random Vibration 50
12 Shock 50
13 Acoustic Level 50
14 Schedule of Development 52

INTRODUCTION

Japanese scientists are now about to start their interplanetary/planetary missions by their own spacecraft and launcher.

The capability of the Mu launcher will be highly improved in coming several years. And many discussions on possible mission for the interplanetary physics have been done in the scientific symposiums held by the Institute of Space and Aeronautical Science.

The above improvement for launching capability will have been feasible by 1984. Although the capability of the launcher is not so sufficient, it can launch a spacecraft into the interplanetary space to carry out some physical experiments.

Through such experiments, it is expected to accumulate experiences on the interplanetary experiment and also to contribute to the international relationships. It is desirable for the Japanese scientists to join to the international cooperation more actively than present.

In the beginning of 1986, Halley's Comet will again approach the sun after its 76 years' recurrent trip in the solar system. This apparition is the first chance to carry out the international scientific observation of the comet by means of various kinds of modern and sophisticated space technology.

The Institute of Space and Aeronautical Science is now planning to send a scientific space probe, tentatively called Planet-A, to participate in this great event by taking vacuum ultra-violet images of the comet and by measuring the solar wind plasma near the comet.

The Institute is now developing an improved type of Mu satellite launcher which will be able to launch Planet-A spacecraft of 135 kg in weight into a flyby orbit near Halley after its perihelion passage.

Following the normal development plan in the Institute, a test spacecraft will be launched into the interplanetary space to prove the launching capability of the new launch vehicle, new techniques such as velocity and attitude controls and deep space communication capabilities. The spacecraft is now tentatively designated as MS-T5.

In this spacecraft, some scientific payloads will also be installed for the study of the solar wind.

This paper presents a general description of the Planet-A mission as well as that of the MS-T5 mission.

SCIENTIFIC OBJECTIVES

Comet is the fossil of the primitive solar nebula, a disc-shaped complex of gas and dust grain particles around the primitive sun at the formation time of the planetary system.

Since comets contain volatiles, they may be more primordial than the meteorites. The observation of these objects will give clues to determine the composition of gas in the solar nebula except H_2 and He.

Some investigators are expecting that comets may be correlated with the origin of life on the primitive earth. Consequently the study of comets is important not only in the exploration of the origin of the solar system, but also in the biological sciences.

Halley's Comet is probably the most active short period comet ever observed by human being. It has many records of apparition from very ancient time with a 76 years period.

The last apparition was in 1910 and was scientifically studied by many telescopes from the ground. In the next 1985-86 apparition, the perihelion passage of the comet will take place of Feb. 9, 1986 at the opposite side of the sun relative to the earth.

Therefore, it will be difficult to observe the comet in its most active phase from the ground. By taking into accounts such an orbital condition of the comet it is quite important to observe the comet from other place than on the earth to study the variation of the comet activity.

One of the measures of the cometary activity is the sublimation of volatile materials from the suface of the nucleus. H_2O is the main volatile material and may start to sublimate at the distance of $1 \sim 2$ AU from the sun.

The sublimated H_2O will be photodissociated to OH+H and successively to O+2H. The photodissociated hydrogen will form a large hydrogen cloud around the nucleus. The hydrogen cloud can be seen by the hydrogen Lyman alpha line through the VUV imaging on board the Planet-A spacecraft.

A successive series of Lyman alpha pictures of the hydrogen coma around Halley's Comet during several tens days before and after perihelion will give us the scientific information on the growing and decaying processes of the hydrogen coma.

The velocity of expanding hydrogen atoms is strongly dependent on the formation mechanism of the atom. Direct photo-dissociation of water results in the atomic velocity of 20 km/s, but the atoms formed by the dissociative recombination of H_3O^+ with electrons or by the photo-dissociation of OH radicals have a velocity of 8 km/s. The velocities so far observed have been very scattered and a long time observation of the hydrogen coma may give a realistic answer to this problem.

The European mission to Halley's Comet, Giotto, is planned to carry out in situ measurements like mass-spectrometry and also imaging in visible region as well as spectrometry. The coordinated studies of Planet-A with this mission can be expected to give good scientific return to the cometary science.

The interaction of the comet with the solar wind is especially strong in its plasma tail. If the shock front accompanied by the solar flares propagates in the interplanetary space, it may cause a turbulent structure in the cometary coma as well as in the plasma tail.

Therefore, it is very important to measure the solar wind in the vicinity of the comet simultaneously by many probes in the same solar wind sector. From this point of view, Planet-A will carry out a solar wind measurement to investigate the flux intensity, energy spectrum as well as the angular distribution in the ecliptic plane.

During the cruise of Planet-A, the above solar wind measurement will be the main item of the mission and will cooperate with the experiments onboard MS-T5.

The MS-T5 spacecraft is the test satellites, to prove the capability of the newly developed satellite launcher, new techniques such as attitude and velocity control system and deep space communication, etc.
However, some scientific experiments are being installed to study macroscopic and microscopic structures of the solar wind.

They are the magnetic field measurement, plasma wave measurement and solar wind particle measurement.
The solar wind is a turbulent plasma flow with magnetic fields in it. Therefore, the first Japanese Interplanetary spacecraft carries these minimum payload to study structure of the solar wind.

The coordination of these two spacecraft is now being planned. MS-T5 will be launched about 8 months before Planet-A launching, and is planned to be located in the downstream of solar wind in the same sector as Planet-A will locate near Halley's Comet at the fly-by time. By such a coordinated observation, the condition of solar wind which affects the comet's activity can be more completely understood.

ORBIT DESIGN

According to the primary scientific objective of Planet-A, that is, observation of Halley's Comet by UV camera, and considering the limited capability of the launch vehicle, requirements on the trajectory of Planet-A are rather loose.

At this stage, it is requested to observe Halley's Comet at its post-perihelion phase, where its activity level is expected to be highest, in as close as possible proximity to the comet.

As a result of preliminary studies, two launch windows were found, one in early 1985, and the other in the summer of 1985. Fortunately both are compatible with the fisherman regulation which permits launchings only in the summer and the winter season. Then, Planet-A is to be launched in the summer launch window, while the earlier window is so allotted to MS-T5 that the distance between two spacecraft is to be very short when Planet-A encounters Halley. This situation is desirable for scientific objectives as mentioned above.

HELIOCENTRIC TRAJECTORY

Planet-A

Planet-A departs from the earth on August 14, 1985, and encounters Halley's Comet on March 8, 1986 near its descending node. The trajectory lies almost three fourth revolution around the sun. Fig. 1 shows the heliocentric geometry of the mission, and Table 1 lists the relevant parameters of the nominal trajectory. Range between the earth and the spacecraft and the sun-earth-spacecraft angle are shown in Fig. 2

20 days of launch window until the beginning of September is available, during which launching with C_3 below 9 km^2/sec^2 is possible.

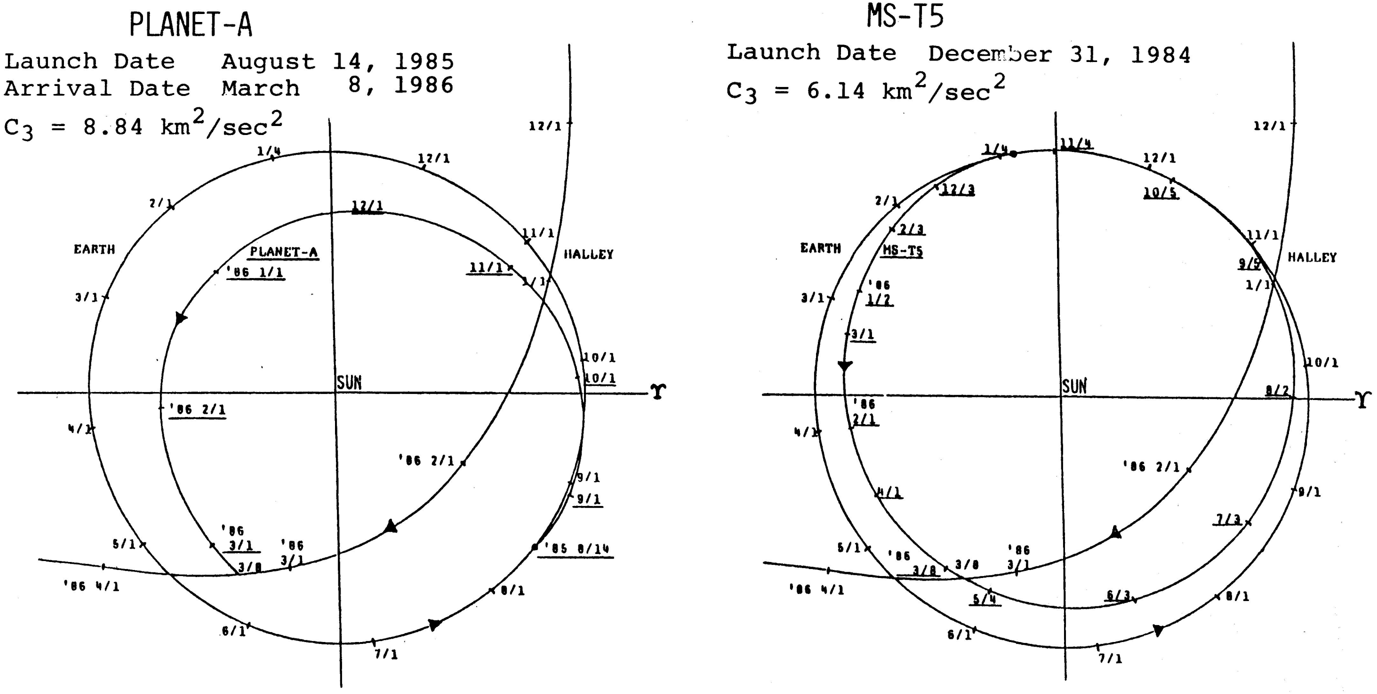
While close encounter is not necessarily required, an encounter trajectory has been selected as nominal. Considering that the launch vehicle itself is under development now, in case the spacecraft grows out of its launch capability, it is possible to modify the nominal trajectory to the one with lower launch energy at the expense of close proximity to the Comet. It is estimated that 1 km^2/sec^2 reduction in C3 results in about 3.5 kg increase in the spacecraft weight.

MS-T5

MS-T5 is to be launched on December 31, 1984 which means just a new year day of 1985 in JST. As shown in Fig. 1, MS-T5 goes one and a half around the sun, and when Planet-A encounters Halley the location of MS-T5 is close to them (about 0.1 AU).

Table 1 Nominal Orbital Parameters of PLANET-A and MS-T5

	PLANET-A	MS-T5
Launch Date	August 14, 1985	December 31, 1984
Arrival Date	March 8, 1986	-------
a (km)	127,252,082.	136,235,370.
e	0.193	0.102
i (deg)	0.699	1.314
Ω (deg)	140.93	280.11
$\tilde{\omega}$ (deg)	147.15	244.67
τ	January 10, 1986	May 1, 1985
C_3 $(km/s)^2$	8.84	6.14


PLANET-A

Launch Date August 14, 1985
Arrival Date March 8, 1986
$C_3 = 8.84\ km^2/sec^2$

MS-T5

Launch Date December 31, 1984
$C_3 = 6.14\ km^2/sec^2$

Fig. 1 Heliocentric Orbit of PLANET-A and MS-T5

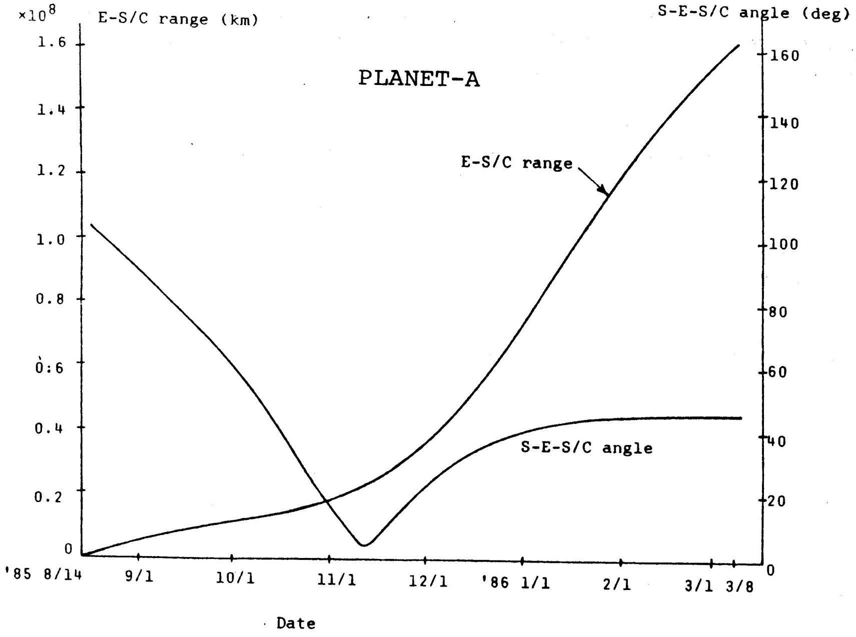


Fig. 2 Range between the Earth and the Spacecraft and the Sun-Earth-Spacecraft Angle (PLANET-A)

The launch energy C_3 for MS-T5 is about 2.7 km^2/sec^2 less than for Planet-A as shown in Table 1.

GEOCENTRIC TRAJECTORY

Planet-A will be launched easterly from Kagoshima Space Center (KSC) located at the southern part of Japan (31.25°N, 131.08°E). Direct injection without parking is employed to save the weight of the attitude control system which is required to reorient the kick stage on the parking orbit, that is, the spin-stabilized third stage and kick stage are fired serially near the apogee of the second stage trajectory.

This, with the constraint on the launch azimuth from range safety consideration, leads to the limitation of the launch asymptote declination. It should be below about -20° to keep the injection flight path angle reasonably low ($< 5°$) to avoid the gravity loss during the injection phase.

The launching scheme of MS-T5 is almost as same as that of Planet-A. The injection parameters of those spacecraft are listed in Table 2 while the fully optimized ones are under study.

Table 2 Injection Parameters of PLANET-A and MS-T5

	PLANET-A	MS-T5
Launch Time (UT)	0h15m Aug. 14, '85	19h55m Dec. 31 '84
Injection Altitude (km)	200.0	200.0
Latitude of Injection Point (deg)	30.90	30.92
Longitude of Injection Point (deg)	138.00	138.00
Absolute Burnout Velocity (km/s)	11.366	11.246
Flight Path Angle (deg)	2.32	1.79
Azimuth Angle (deg)	94.89	94.81

LAUNCH VEHICLE

The Planet-A spacecraft is launched by the Mu-3U rocket whose cutaway view and specification are shown in Fig.3 and Table 3 , respectively. The Mu-3U is now under development by ISAS as the newest version of the Mu family launch vehicles. Its total length and maximum diameter (excluding SOBs) are 28.2 m and 1.65 m, respectively. The launch weight will be 61 tons. It can place a 700-kg-class payload into low earth orbit.

The first stage consists of a polybutadiene solid propellant motor with 27.1 tons of propellant, a quadruply segmented maraging steel motor case, and a fiber reinforced plastic nozzle. The internal burning motor gives an average thrust of 1.120×10^3 newtons at sea level condition for an effective burning duration of 56 seconds.

Attached to the first stage are twin polybutadiene SOBs, each of which develops an average thrust of 298×10^3 newtons for a duration of about 31 seconds. Eight proportional type TVC (Thrust Vector Control) valves are accommodated around the nozzle of the main engine.

The second stage utilizes a maraging steel motor case and has 9.9 tons of polybutadiene propellant delivering an average thrust of 519×10^3 newtons in vacuum for a duration of 52 seconds. The second stage control system consists of attitude control electronics, TVC with on-off valves and four hydrazine mono-propellant side jet modules.

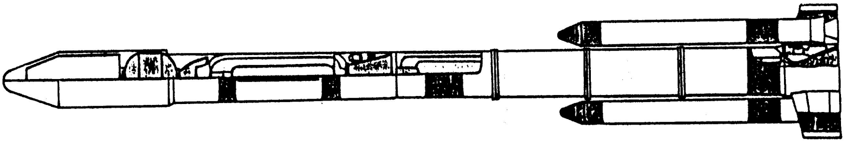


Fig. 3 Cutaway View of Mu-3U

The third stage consists of 3.3 tons of polybutadiene propellant, titanium alloy motor case, and highly submerged fiber reinforced plastic nozzle, It gives an average thrust of 118×10^3 newtons for about 82 seconds. The third stage is spin stabilized.

To inject the Planet-A spacecraft, the fourth stage motor with about 400 kg propellant will be optionally added.

SPACECRAFT

DESIGN PHILOSOPHY

The objective of the Planet-A is to conduct scientific investigations of Halley's Comet and interplanetary plasma measurements en route using spin-stabilized spacecraft launched by the Mu-3U vehicle.

The mission of photographing of the Comet in hydrogen emission (Lyman alpha 1216Å) prefers that the camera should be mounted on a non-spin platform. However 3 axis controlled spacecraft was avoided to reduce the complexity of thermal control. Then the camera must be installed on a spinning body of which spin rate is very low. While a high gain antenna should be oriented to the earth in order to maintain radio communication with the earth station which is to be constructed in Japan.

Thus the spacecraft system was determined as follows.

Table 3 MU-3U Specification

SPECIFICATION	FIRST STAGE	SOB	SECOND STAGE	THIRD STAGE
THRUST (kN)	1,120 (SEA LEVEL)	298 (SEA LEVEL) x 2	519 (VACUUM)	118 (VACUUM)
PROPELLANTS	SOLID	SOLID	SOLID	SOLID
SPECIFIC IMPULSE (s)	236 (SEA LEVEL)	236 (SEA LEVEL)	280 (VACUUM)	292 (VACUUM)
EFFECTIVE FIRING DURATION (s)	56	31	52	82
PROPELLANT WEIGHT (t)	27.1	4.00 x 2	9.85	3.30
STAGE WEIGHT (t)	34.6	5.00 x 2	11.9	3.70
GUIDANCE	RADIO-INERTIAL GUIDANCE			
DIAMETER (m)	1.65			
TOTAL LENGTH (m)	28.2			
LAUNCH WEIGHT (t)	61.0			

The spacecraft is spun with 5 rpm during its cruising phase, and a high gain antenna is so fully despun that down link communication is maintained.

Before taking photographs, spin is reduced to 0.2∿ 0.5 rpm by running up a bias momentum wheel to stabilize the spacecraft attitude. During this phase, only the up link communication is possible and imaging data stored in a data recorder using magnetic bubble memory are transmitted to the earth after the spin rate up to 5 rpm again.

During both phases, the spin axis is controlled to be perpendicular to the ecliptic plane.

The spacecraft will weigh about 135 kg and will carry between 10 and 15 kg of scientific instruments. The program uses state-of-the-art components, subsystems, and fabrication techniques to minimize costs and development time.

The main mission of MS-T5, which is to be launched prior to Planet-A, is the certification of Japanese first interplanetary spacecraft, though it will also carry out scientific tasks such as measurements of plasma wave and interplanetary magnetic field. Therefore the design of MS-T5 is almost same as Planet-A except that it has 4 booms for scientific instruments.

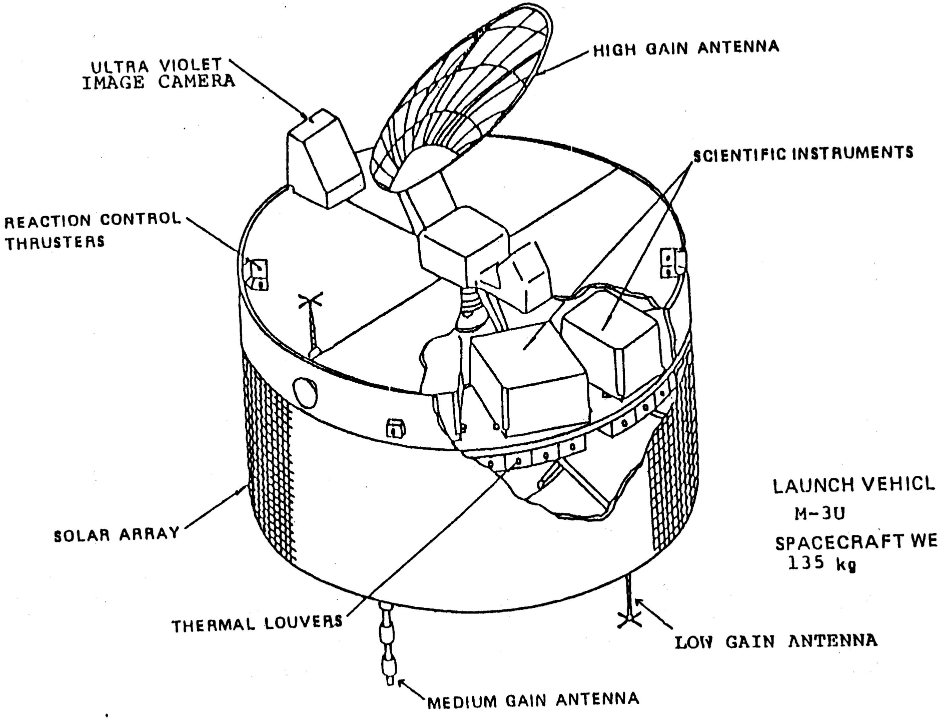
WEIGHT AND CONFIGURATION

Fig. 4 shows a cutaway illustration of the spacecraft. Basically the spacecraft has a cylindrical shape which is 1400 mm in diameter and 700 mm in height with the high gain mechanically despun antenna attached to the upper end of the spacecraft. The roll-to-transverse inertia ratio is expected to be about 1.3.

Fig. 5 is a block diagram of the spacecraft system.

Table 4 lists the spacecraft weight beakdown for Planet-A. MS-T5 spacecraft is about 5 kg heavier than Planet-A only because of the difference of scientific instruments.

Fig. 4 *PLANET-A Spacecraft*



LAUNCH VEHICLE:
M-3U
SPACECRAFT WEIGHT:
135 kg

Fig. 5 **PLANET-A** SYSTEM BLOCK DIAGRAM

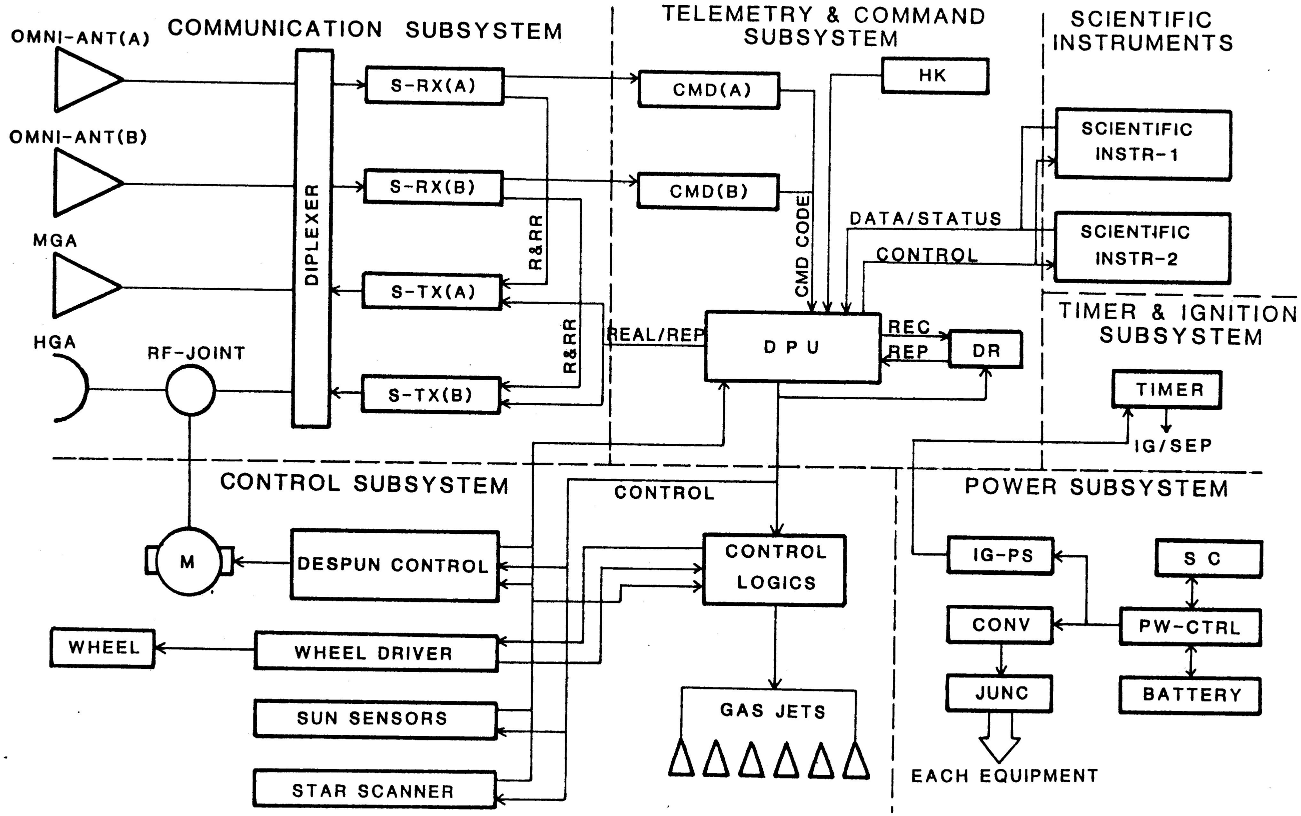


Table 4 PLANET-A Weight Breakdown

Power Subsystem	
Solar Cell Panel	12.3
Battery	2.2
Converter	5.5
Power Control Unit	1.4
Ignition Power Unit	0.2
Circuits	0.8
Sub-Total	22.4
Communication & Antennas Subsystem	
Low Gain Antennas	1.2
Medium Gain Antenna	1.0
High Gain Antenna	3.3
Transmitter & Receiver	5.6
Rotary Joint	1.5
Antenna Support	1.2
Despun Motor	6.3
Motor Control Unit	1.5
Sub-Total	21.6
Telemetry & Command Subsystem	
Data Processing Unit	5.7
Command Decoder	3.0
Data Recorder	3.2
House Keeping	1.0
Timer	1.2
Sub-Total	14.1
Control Subsystem	
Momentum Wheel	8.6
Wheel Driver	1.4
Control Circuits	3.0
Sun Sensors	0.8
Star Scanner	1.4
Nutation Damper	0.8
Propulsion System	15.0
Sub-Total	31.0
Structure & Thermal Subsystem	
Structure	17.9
Thermal Louvers	5.0
Blankets etc.	3.0
Sub-Total	25.9
Integration	
Bolts etc.	4.5
Circuits	6.0
Sub-Total	10.5
Scientific Instruments	10.0
Total Weight	135.5 (kg)

SCIENTIFIC INSTRUMENTS

Lyman-α Imaging for Halley's Comet

The hydrogen coma is originated from the photodissociation of water vapour released from the water ice which is a major constituent of volatile components in the comet nucleus. The present experiment is aimed to take the image of hydrogen coma of Halley's Comet by means of hydrogen Lyman-α imaging device.

The device is composed of the image forming part (vacuum ultraviolet telescopic mirror lens), image detector (image-intensifier and solid-state image-pickup (CCD)) and controlling electronics including a micro-processor.

The hydrogen coma is illuminated by resonant fluorescent Lyman-α emission as well as the background interplanetary hydrogen glow. The detection limit of 1kR is, therefore, an appropriate threshold of observation in sufficient S/N ratio. This requirement of low light level imaging is difficult to achieve due to the spacecraft spinning adopted on Planet-A for its attitude control.

In order to prevent the image blur associated with spacecraft spinning, the electronic charge on CCD which is corresponding to the objective image is shifted synchronizing with the shift of image on CCD caused by the spinning. Then, it is possible to integrate the electronic charge corresponding with object image in CCD up to about 1 second by this spinning synchronized charge shift.

The image data are transmitted to the earth station with effectively about 50 bps transmission rate. Therefore, the image data (size: about 150×250 pxls.) are subjected to data compression to retain the time resolution of the imaging.

The image data read out from CCD are converted into digital form and stored temporarily in the buffer memory. The microprocessor reads out the image data from the buffer memory and stores them in the compressed form into the main memory with the housekeeping data.

Then next imaging is carried out by the pre-set command sent out from the on-board RAM memory. After the execution of pre-set instruction sequence in the RAM, all data stored in the main memory are sent back to the ground station. The received data are converted into the original form through the inverse process to on-board compression one.

Fig.6 shows schematically the camera subsystem.

Solar Wind Measurement Onboard Planet-A

The objective of the experiment is to measure three-dimensional distribution of the solar wind plasma within ± 30° to the ecliptic plane. The experiment consists of two charged particle detectors for electrons and ions respectively, in the energy range from 30 eV up to 16 keV in 96 steps.

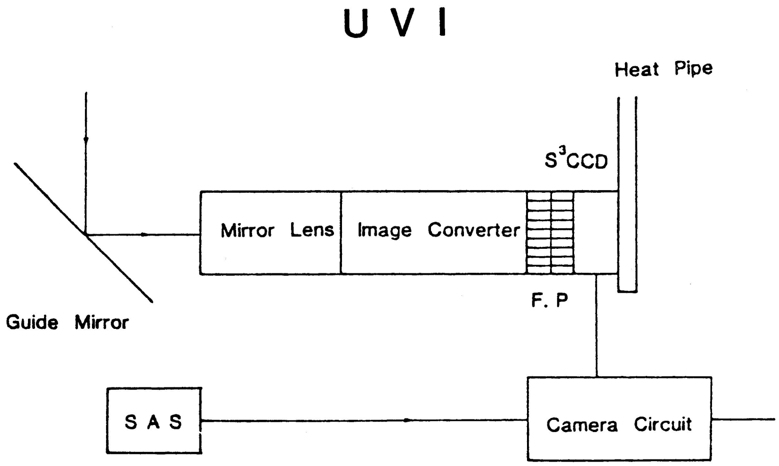


Fig. 6 UV Camera Block Diagram

The energy En of n-th step is represented by the equation, $E_n=30\cdot\exp[0.066(n-1)]$. Four energy-scanning modes are prepared; 96 steps over the whole energy range, 64 steps below 1920 eV, 64 steps above 250 eV, and 48 steps in the whole energy range.

Each detector is a 270° spherical electrostatic analyzer with a micro-channel plate (MCP). The view direction is perpendicular to the spin axis of the spacecraft, with a view field of ± 2.5° in the azimuthal direction and of ± 30° in the polar direction.

The anode of MCP is split into five in order to measure a 5-points angular distribution with a resolution of 12° in the polar direction of ± 30°. In the azimuthal direction, the angular distribution is obtained by using the satellite spin, with 5.625° interval within ± 22.5° to the solar direction and of 22.5° interval in other directions.

During a spin, four energy steps are scanned periodically in every interval (5.625°/22.5°); that is, the step is changed every spin angle of 1.40625° or 5.625°.

The experiment is operated only at high spin rate (∿5 rpm, nominal). At high bit rate (1024 bps), observations of three-dimensional distribution of electrons and ions are carried out continuously.

The data during one spin are stored temporarily in the buffer memory and then transmitted in 10 PCM frames (10 seconds) during the next interval. It takes 24-spin period of time to complete the whole sequence of 96 energy steps.

At low bit rate (64 bps), observation of two-dimensional distributions is carried out intermittently every 512 seconds. The data during 16-spin periods (64 energy steps) are stored temporarily in the buffer memory and then transmitted in 32 PCM frames (512 seconds).

Plasma Wave Probe Onboard MS-T5

The detection of the plasma wave instability at f_p (plasma frequency) and the electron cyclotron harmonic wave emissions are planned to detect in a relatively wide frequency range from 50 Hz to 200 kHz.

The principal purpose of this observation is to clarify the spectra of the plasma waves combined with the magnetic field and solar wind data, that largely control the effective viscosity and the effective collisions of the interplanetary space plasma.

The generation of the type III radio waves through the interplanetary space due to the high energetic particle stream ejected by the solar flare is also intended to observe as a byproduct of this plasma wave observation.

The electric field is planned to be observed in a frequency range from 4 kHz to 200 kHz with the bandwidth of 2 kHz, the center frequency of the frequency spectrum analyser is swept in 8 sec from 4 kHz to 200 kHz. A short dipole antenna with length of 10 m (T.B.D.) from tip to tip is employed for this purpose to realize the threshold value of 0.01 volt for the input impedance of 75 Ω.

The magnetic field component of the electromagnetic wave is planned to measure in a frequency range from 50 Hz to 5 kHz with the bandwidth of 20 Hz; the center frequency of the spectrum analyzer is swept in 8 sec to cover the whole frequency range. A search coil is set as the sensor to pick up the oscillating component of the magnetic field above the thereshold of 5pT.

The block diagram of the total system is indicated in Fig. 7. The total planning weight of the system is 6.7 kg including the sensor. The data are sent by PCM telemetry system; the sampling rate of the plasma wave is 48 words per every 8 sec.

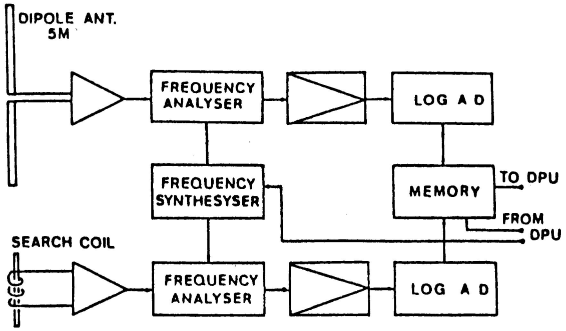


Fig. 7 Plasma Wave Probe Block Diagram

Solar Wind Measurements Onboard MS-T5

Solar Wind plasma parameters are measured by means of retarding potential trap. The parameters to be measured are ion temperature, bulk velocity, plasma density and electron temperature.

Two sensors are used; one is for electrons and the other is for ions. The sensors consists of 4 plane grids (G1, G2, G3, G4) and a collector plate with diameter of 120 mm.

The sensors are mounted on the outer wall of spacecraft in such a way that grid G1 is part of the spacecraft skin. 16 different retarding voltages are stepwisely applied to the 2nd grid (G2) and the retarding voltage is kept constant during one spacecraft spinning.

Swing range of the retarding voltage is for ion measurement from 300 V to 2.5 kV (which corresponds to 700 km/sec bulk velocity). The retarding voltage is modulated by 2 kHz square wave to remove photoelectron effect. G3 is used to reduce the capacitive coupling between G2 and collector. G4 reduces the photoelectron current to the collector.

2 kHz current component is detected by means of lock-in amplifier. DC current is also detected. The current to be detected is from 10^{-13} A to 10^{-9} A. Swing range of the retarding voltage for electron measurement is from 0 to 100 V.

30 data points are sampled in one spacecraft spin with sunward synchronization. Ion and electron currents are both first stored in memory and transmitted to the ground when sensors point anti-solar direction.

Angular acceptance of the detector is established by a collimator that limits the acceptance angle of ± 20° in the azimuthal direction but allows particles to enter over an angular range of roughly ± 60° in the polar direction.

Total weight is estimated as 3.2 kg.

Measurement of the Interplanetary Magnetic Field Onboard MS-T5

In order to observe macroscopic and microscopic structures of the interplanetary magnetic field, a ring-core type fluxgate magnetometer is installed on the spacecraft.

The three orthogonal ring-cores with a diameter of 25.4 mm are packed in a case with a size of 90ϕ×150 mm, which is fixed on the tip of the boom with 4 m (T.B.D.) length. The boom is extended after the injection of the spacecraft into the interplanetary space.

The dynamic range and the noise envelope are 64nT and < 30 pT, respectively. The resolution is expected to be ± 62.5 pT/bit and ± 250pT/bit for the far-space and the near-space, respectively. The drive current for the cores has a frequency of 12.5 kHz. The size of the onboard electronics is to be 170× 160×100 mm.

The net weight including the electronics, interface, case, cables, and sensor (excluding the boom and its extension system) is planned to be 3.9 kg. The power consumption is 5 W.

As for the data transmission 10 bit/comp and 8 bits/comp are used for the near-space and the far-space respectively.

DATA PROCESSOR

The data processor of the Planet-A consists of five units, which are timing control unit, data formatting unit, coded PCM transmission unit, control unit, and picture data compression unit. These units and their interconnection are schematically shown in Fig. 8.

Timing Control Unit

By frequency division of a master crystal oscillator, formatting pulse, timing pulse, and clock signal are generated in this unit to control the subsystems of the data processor and magnetic bubble data recorder.

Data Formatting Unit

In this unit the analog and digital data from other onboard instruments are multiplexed into a frame format together with the synchronizing pattern and clock signals. The multiplexed data are provided to the coded PCM transmission unit or the magnetic bubble data recorder.

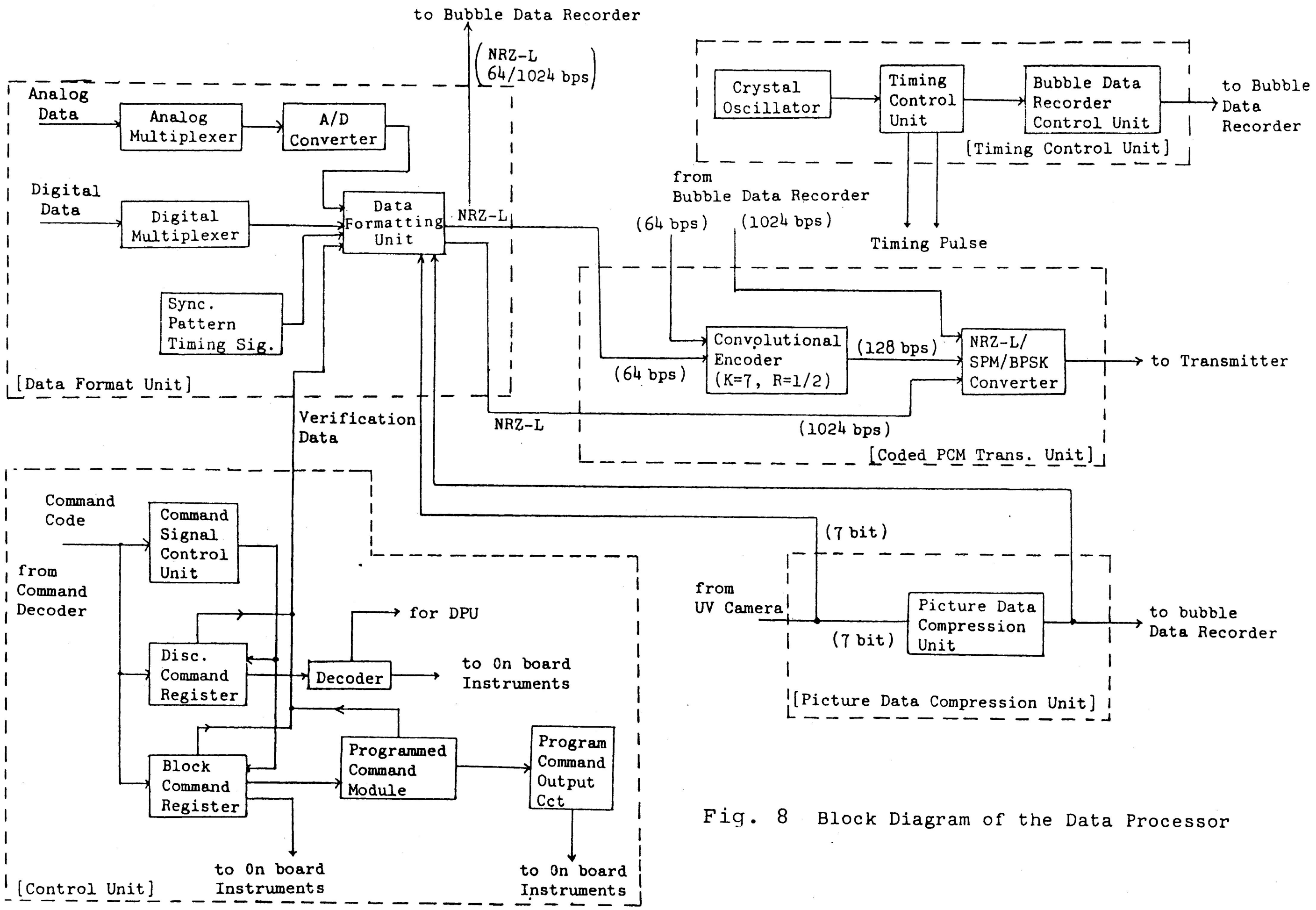


Fig. 8 Block Diagram of the Data Processor

Coded PCM Transmission Unit

In this unit real time signal with bit rate of 1,024 bps arranged by the data formatting unit is sent to the transmitter through a NRZ-L/SPM/BPSK converter, and the signal with bit rate of 64 bps is converted into a convolutional code (K=7, R=1/2), and is sent to the transmitter through a NRZ-L/SPM/BPSK converter.
The stored data in the magnetic bubble data recorder are reproduced in bit rate of 1,024 bps or 64 bps, and are sent to the transmitter after being processed similarly to the case of real time data.

Control Unit

This unit supplies both discrete and block commands sent from the ground station to the instruments to be controlled. This unit also generates command signals arranged by the programmed control device to the instruments in the spacecraft.

Picture Data Compression Unit

This unit compresses the picture data of Halley's Comet taken by the UV camera. The compression is made by the use of both variable words DPCM and sampled data transmission systems.

Main parameters of the data processor are shown in Table 5.

COMMUNICATION SYSTEM

Planet-A communication system consists of telemetry, command and ranging subsystem with S-band transponder and antenna subsystem characterized as follows:

1) Telemetry Subsystem
   a) Convolutionally Encoded PCM (R=1/2, K=7, Q=8)
   b) Data Rate : 1024 bps without coding and 64 bps with coding
   c) DSN Parametric Amp. 30°K
   d) Modulation Waveform : PCM-PSK-PM (Square wave)

Table 5 Parameters of Data Processor

(1) PCM Data Format		5 Kinds
	1 Word	8 Bits
	1 Minor Frame	128 Words
	1 Major Frame	256 Minor Frames
	Bit Rate	1024 BPS (1 Frame/S) 64 BPS* (1 Frame/16S) *Simbol Rate: 128 BPS
	Frame Sync.	3 Words
(2) PCM Modulation		NRZ-L/SPM/BPSK (for 1,024 BPS) NRZ-L/Conv. Code/SPM/BPSK (for 64 BPS)
(3) A/D Converter		Digital 8 Bits Linear Coding Sequential Comparison Conv. Time 138 μs Input Vtg: 0~3 V Input Freq.: 0 < 4.5 Hz Prec.: ±0.8% Full Scale Input Imp.: 200 KΩ
(4) Input Level	Analog:	0~3 V
	Digital:	"1" 4.4~5.5 V "0" 0 ~0.1 V
(5) Output Level	Digital	"1" 4.4~5.5 V "0" 0 ~0.1 V
(6) Bubble Data Recorder Interface	Record Signal	1024 BPS 64 BPS } NRZ-L
	Reprod. Signal	1024 BPS 64 BPS } NRZ-L
(7) Error Correcting Code		Convolutional Code (Viterbi Decoding) K=7 R=1/2 Max. Free Distance, Noncatastrophic, Nonsystematic Code "171", "131" Octal
(8) Discrete Command (D.C.)		224 Items
(9) Block Command (B.C.)		8 Bit Serial Data/Item
(10) Programmable Command		254 Items
	Time Interval	128S/2048S/4096S
	Error Correcting Code at the Execution	Hamming Code Information 8 Bit Check 4 Bit
(11) Picture Data Compression		System: DPCM + Sampled Data Transmission DPCM Coding: Input PCM 7 Bits/Sample Quantizing Level: 15 Variable Word Length: One of the 1, 3, 4, 5, 6 or 8 Bits Compression Rate: 1/2 1/128
(12) Power Consumption		3 W.
(13) Weight		6.8 Kg
(14) Size		340 mm × 250 mm × 210 mm (H)

2) Command Subsystem
   a) Modulation Waveform : PCM(PN)-PSK-PM (Square wave)
   b) Bit Rate : 16 bps with 512 Hz Subcarrier
   c) Command Channel
      Discrete Command : 225 channels(max)
      Block Command : 8 bit/channel
      Programmed Command : 225 channels (max)

3) Ranging Subsystem (TBD)
   a) PN code system
   b) Chopper Frequency : $f_{TX}/4096$
   c) Code Component : 20

The detailed communication system parameters are shown in Table 6 and the link margin vs slant range for each system is shown in Fig. 9 through Fig. 12.

RF SUBSYSTEM

The RF subsystem consists of an S-band transmitter/receiver that performs the functions of a double-conversion superheterodyne command receiver, a turn-around ranging transponder, and a telemetry transmitter. Co-axial switches and diplexers included in the subsystem achieve the proper antenna-mode switching. The transmitters put out either 1 or 5 watts of power on command.

The transmitter frequency is derived in either of two ways: During periods of noncoherent operation, when no uplink signal is present, the crystal oscillators generate the downlink frequency. When an uplink signal is present, indicating command reception, the receiver's phase-locked oscillator may serve as the frequency source for the downlink. The changeover to the phase-locked oscillator occurs either on ground command, or automatically when the receiver locks to the ground signal. In the phase-lock mode, either two-way Doppler or range-rate tracking is possible.

Table 6 Communication System Parameters

	Parameter	Up Link	Down Link	Remarks
1	Frequency	2113.65MHz	2295.37MHz	tentative 2110∿2120 2290∿2300) MHz
2	Spacecraft Parameter			
(1)	Antenna Gain			
	(a) H.G.A.	21.50 dB	22.50 dB	RHC
	(b) M.G.A.	0.0 dB	0.0 dB	(NOTE 1) ±22°(Linear)
	(c) L.G.A.	-3.0 dB	-3.0 dB	(NOTE 2) ±68°(Linear)
(2)	Transmitter Power	———	5W±1 dB	
(3)	Feeder Losses			
	(a)Coax. Cable (b)SW (c)Filters	3.2 dB	3.8 dB	
	(d)Rotary Joint	0.8 dB	0.8 dB	H.G.A. channel
(4)	RX Noise Figure	3.0 dB	———	
(5)	RX PLL Bandwidth (2BL)	20 Hz	———	
(6)	Transponder Loss			
	(a)Limitter Supression	* -29.36 -6.85 [-0.66]	* -29.36 -6.85 [-0.66]	$T=10\log(\frac{P_r}{P_r+N_0 Bnr})$ * $P_r/N_0=$ 32.40(L.G.A.) 55.91(H.G.A.) [69.58](L.G.A.) Bnr=1.5 MHz
3	DSN Parameter			
(1)	Antenna Gain	61.48 dB [42.4 dB]	62.18 dB [43.2 dB]	64mϕ/η=70% (TBD) [10mϕ]
(2)	Transmitter Power	30KW±1dB [10KW±1dB]	———	
(3)	Feeder Loss	1.0 dB [1.0 dB]	0.4 dB [1.0 dB]	
(4)	RX Noise Temp.	———	30°K [125°K]	

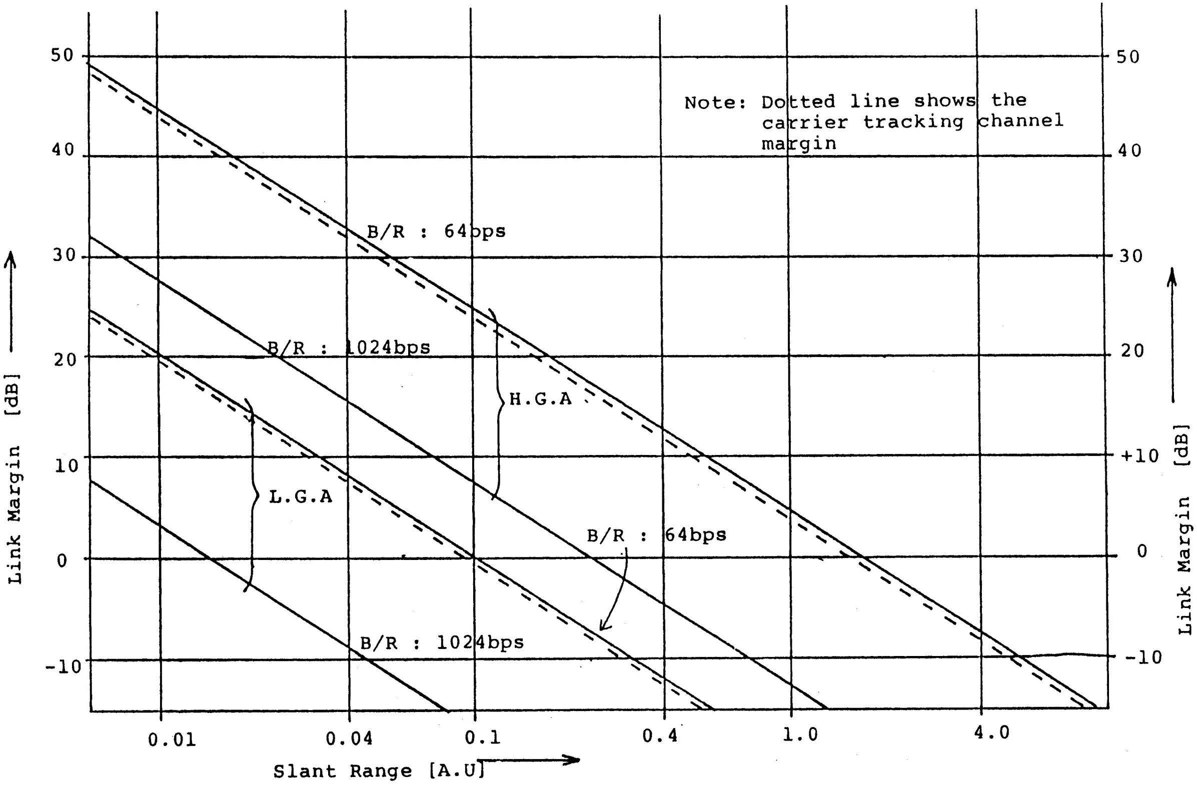
[ ] denotes KSC (Kagoshima Space Center)

Table 6 (continued)

No.	Parameter	Up Link	Down Link	Remarks
(5)	RX PLL Bandwidth	———	10 Hz [100 Hz]	
4	MODEM Parameter			
(1)	Radio Loss	*1.5 dB	*1.5 dB	*Command/Ranging **Telemetry/Ranging
(2)	Telemetry Demod. Loss			
	(a)Predetection Recording	———	1.00 dB	tentative
	(b)Waveform Dist.	———		
	(c)Symbol Synch.	———	1.00 dB	
	(d)Subcarrier Demod.	———		
(3)	Command Demod. Loss			
	(a)Waveform Dist.	0.10 dB	———	tentative
	(b)Bit Synch/Detect.	0.10 dB	———	
	(c)Subcarrier Demod.	0.40 dB	———	
	(d)Circuit	0.10 dB	———	
(4)	Data Bit Error Rate	10^{-5}	10^{-5}	
(5)	Required S/N (Eb/N_0)			
	(a)Telemetry (coded)	———	4.5 dB	R=1/2, K=7, Q=8 Viterbi Decoding
	(b)Telemetry (uncoded)	———	9.6 dB	
	(c)Command	9.6 dB	———	
	(d)Ranging (P_r/N_0) $*T_{(acq)}$	3.92 dB [35 dB]	-06.08 dB [25 dB] (*300sec)	Mu System Threshold 1.869 dB-10log$T_{(acq)}$ ($T_{(acq)} \geqq$ 80sec)
5	Propagation Parameter			
(1)	Slant Range (max)	1.50 AU	1.50 AU	AU: 1.5×10^8 Km
(2)	Axial Ratio (Polarization Loss)	2.0 dB (0.2 dB)	2.0 dB (0.2 dB)	
(3)	Rain Loss	0.2 dB	0.2 dB	
6	System Loss			
(1)	Antenna Pointing Loss (PLANET-A)	0.2 dB	0.2 dB	Pointing Accuracy $\leqq$ ±1.5°
(2)	Antenna Pointing Loss (DSN)	0.2 dB	0.2 dB	
(3)	Antenna Noise Temperature	300°K	20°K [50°K]	EL > 15°

Fig. 9 S/C→DSN Telemetry Link

(Down Link)



Note: Dotted line shows the carrier tracking channel margin

Fig. 10 S/C→KSC Telemetry Link
(Down Link)

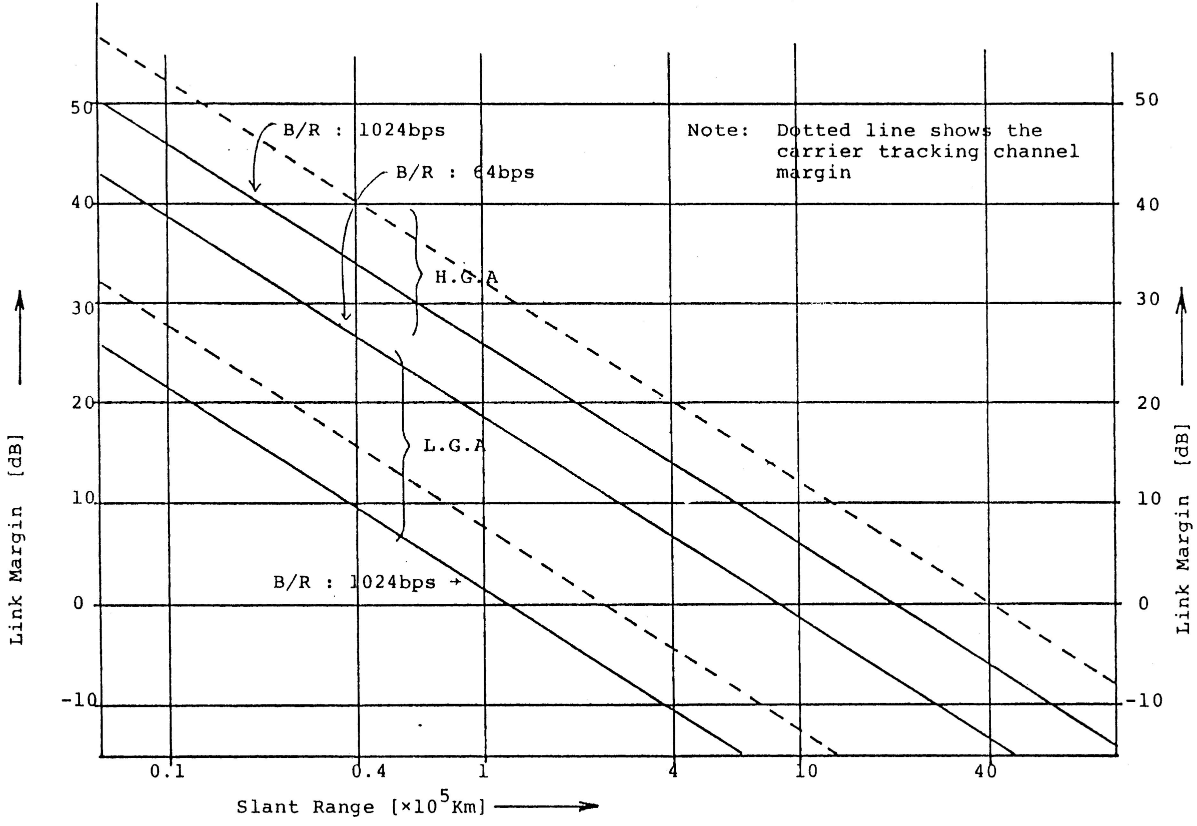


Fig. 11 S/C←DSN Command Link

(Up Link)

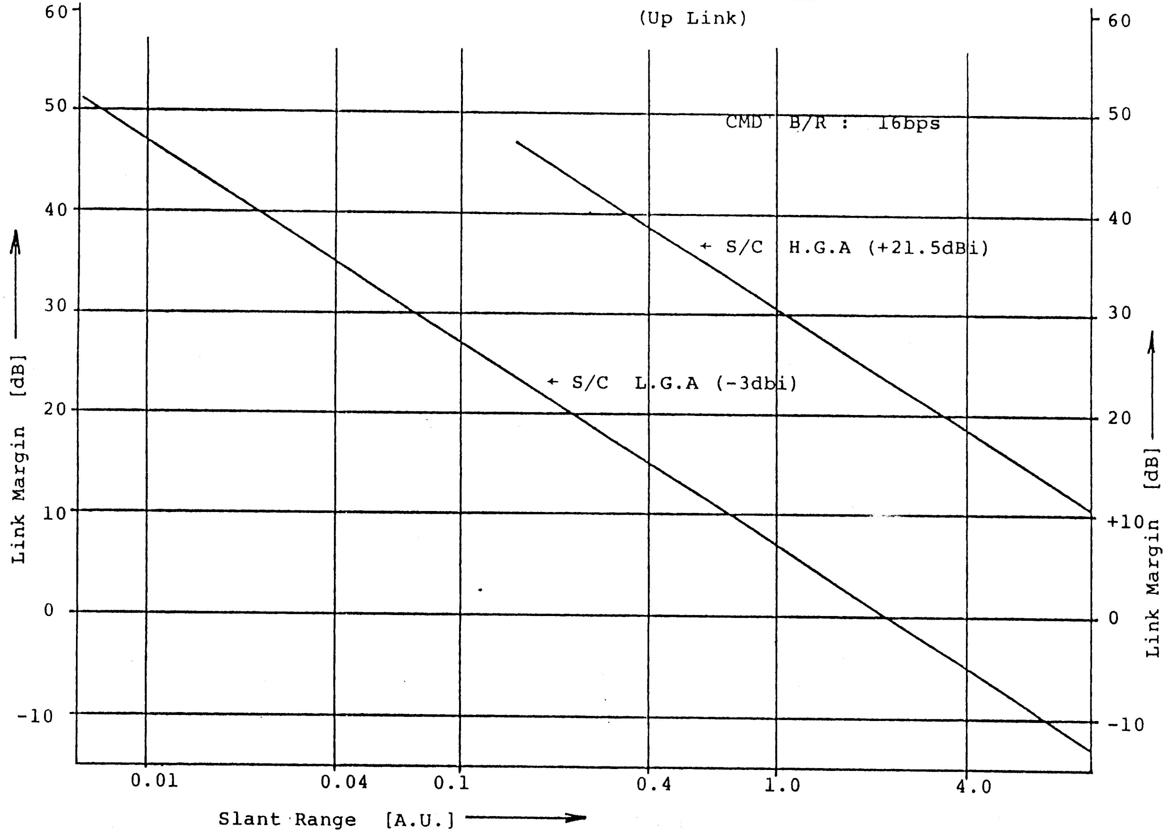
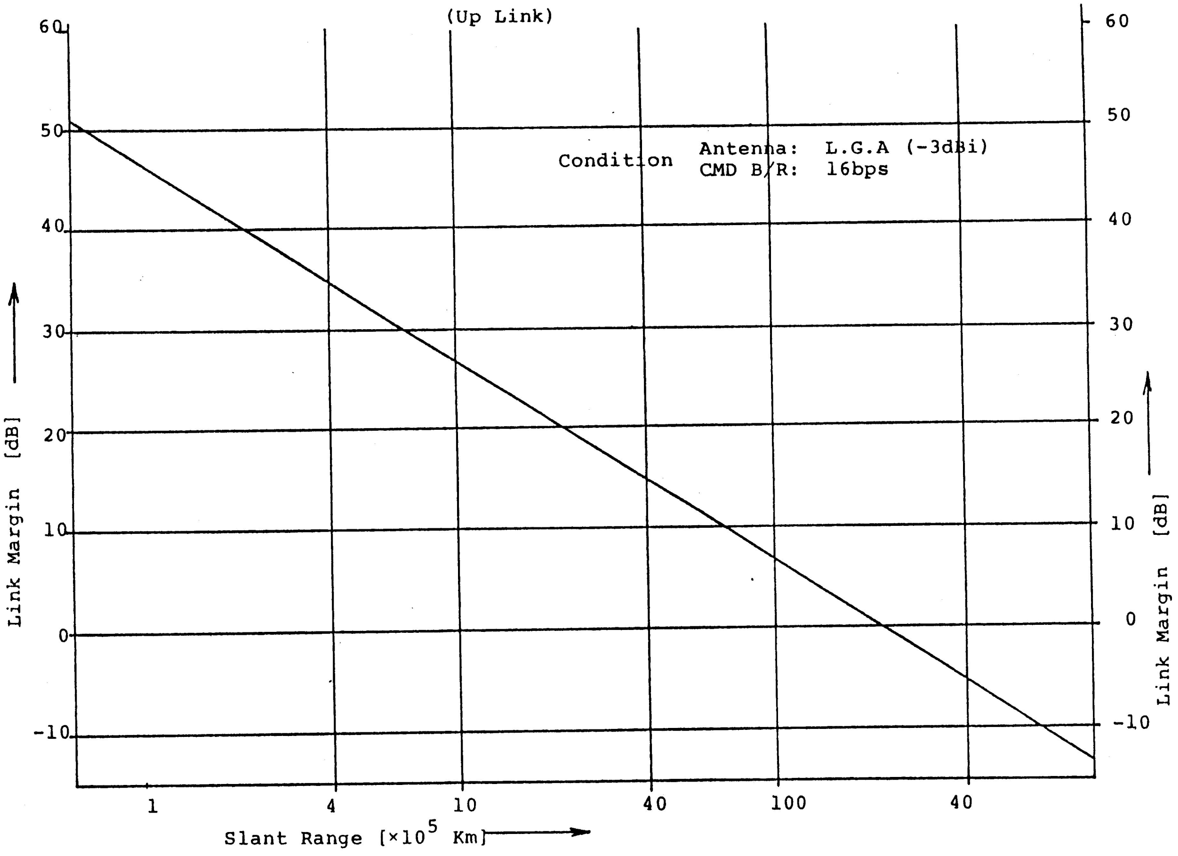


Fig. 12 S/C←KSC Command Link

(Up Link)



ANTENNA SUBSYSTEM

The spacecraft/earth geometry and data rate requirement preclude the use of a single antenna for the total Halley mission.

The antenna system consists of a high gain antenna, a medium gain antenna and a pair of low gain antennas.

The high gain antenna is an offset-parabolic reflector. The diameter is 80 cm and the focal length is 40 cm, and it is fed by a circular polarized horn. This antenna is mounted on a despin motor and is used for telemetry transmission, command reception and ranging in S-band.

The medium gain antenna is a three-element colinear array in S-band and the gain is 6 dB. This is mounted on the aft panel of the spinning spacecraft, and is used for the backup of the high gain antenna.

One of the low gain antenna is mounted on the aft panel besides the medium gain antenna and the other is on the fore panel of the spacecraft. Both low gain antennas are used for telemetry transmission, command reception and ranging when the spacecraft is in the near earth orbit.

The antenna subsystem is characterized as follows:

1. High Gain Antenna

1)	Antenna Type	Offset Parabolic Antenna
2)	Aperture Diameter	80 cm
3)	Frequency Range	2050 MHz ∼ 2350 MHz
4)	Polarization	Right Hand Circular
5)	Gain	Up Link ≧ 22 dBi Down Link ≧ 23 dBi
6)	Half Power Beamwidth	10°
7)	Axial Ratio	≦ 2 dB
8)	Weight	≦ 3.3 Kg

2. Medium Gain Antenna

1)	Antenna Type	3 elements colinear broadside array	
2)	Frequency Range	2050 MHz ∼ 2350 MHz	
3)	Polarization	Linear	
4)	Gain	≧ 6 dBi	in the plane perpendicular to plus spin axis
		≧ 0 dBi	within the direction perpendicular to plus spin axis 22°

5) Pattern	Pancake Beam
6) Weight	≦ 1.0 Kg

3. Low Gain Antenna

1) Antenna Type	Turnstile Antenna or Cross Dipole Antenna	
2) Frequency Range	2050 MHz ~ 2350 MHz	
3) Polarization	Right Hand Circular	
4) Gain	LGA-1 ≧ -8 dBi	over than 90% of the hemisphere directed to aft side of the spacecraft.
	≧ 0 dBi	within aft side spin axis ±60°
	LGA-2 ≧ -8 dBi	over than 70% of the hemisphere directed to forward side of the spacecraft.
5) Pattern	Omni-directional Pattern	
6) Weight	≦ 1.2 Kg	

4. RF Rotary Joint

1) Type	1 channel coaxial type rotary joint
2) Frequency Range	2050 ~ 2350 MHz
3) Handling Power	≦ 5 W
4) VSWR	≦ 1.2
5) Insertion Loss	≦ 0.6 dB
6) Weight	≦ 1.5 Kg

5. Despin Motor Assembly

1) Type	Permanent magnet synchronous blushless DC two-phase Motor
2) Spin Rate	5 rpm or 0.2 rpm (nominal)
3) Pointing Resolution	0.7°
4) Pointing Jitter	0.1°
5) Weight	ADM ≦ 4 Kg
	MDE ≦ 1.2 Kg
	SCE ≦ 1.5 Kg

(ADM: Antenna Drive Mechanism)

(MDE: Motor Drive Electronics)

(SCE: Servo-Control Electronics)

HOUSE KEEPING SUBSYSTEM

The main purpose of the house-keeping (HK) subsystem is the monitoring of temperature at several points in the spacecraft and the oprational conditions of the power subsystem and gas jet subsystem.

The number of items monitored by HK subsystem is 64 including 2 items for calibrating this subsystem itself. Each item is measured every 64 seconds at high transmitting rate and 1024 seconds at low bit rate respectively.

Platinum resistance wire is used as the sensor of temperature. The weight of this subsystem is 1.2 kg and the power consumption is about 1.4 watt.

The block diagram is shown in Fig. 13 .

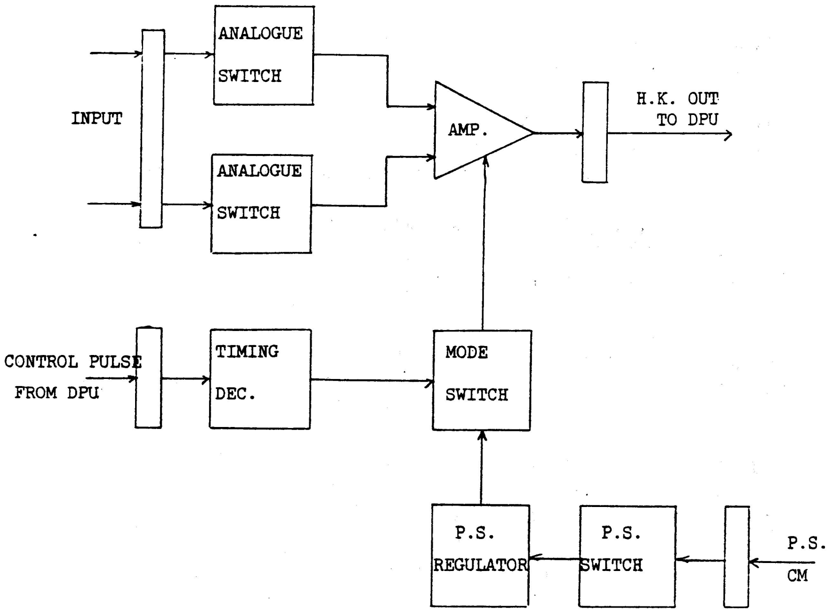


Fig. 13 Block Diagram of HK Subsystem

ATTITUDE AND VELOCITY CONTROL

The control subsystem provides the attitude sensors, bias momentum wheel, elecronics and mono-propellant gas jet engines to accomplish several tasks.

Event sequence after the spacecraft injection may be divided into five phases as shown in Table 7.

The first phase includes re-tracking after 5 hrs invisible time following spacecraft injection, and spin reduction from 120 rpm to 5 rpm for spacecraft operations.

In the second phase, spin axis is oriented to the direction perpendicular to the sun-spacecraft line in order to assure solar power, then precession maneuver makes the attitude of spacecraft normal to the ecliptic plane.

After first 3 days of accurate ranging and orbit determination, a midcourse correction maneuver will be done. Though this maneuver is basically divided into 3 steps, as shown in Table 7, the attitude in which thermal louvers face to the sun is prohibited by reason of spacecraft thermal conditions.

In this case midcourse correction must be done by spin-synchronized pulse jets leaving the attitude normal to the ecliptic.

Phase 4 covers interplanetary cruise up including attitude maintenance at about 10 days interval.

The final phase includes a bias monentum wheel run-up and wheel unloading for the purpose of stabilization of the spacecraft attitude during photograhs of the Comet are taken. During this phase the spin rate of spacecraft itself is reduced to 0.2-0.5 rpm.

Sensors and Wheel

Sun sensor

The spin type sun sensor provides the sun-line to spin axis angle and the sun pulse information. It

Table 7 Event Sequence

Phase	Event	Event Magnitude	Spin Rate after Event (rpm)
1	Spacecraft Injection		120
	Despin	-115rpm	5
2	Orient to Normal-to-Sun	60deg	5
	Orient to Normal-to Ecliptic	180deg	5
3	(Orient for Midcourse Correction	180deg	5)
	Midcourse Correction	30m/s	5
	(Re-orient to Cruise	180deg	5)
4	Attitude Maintenance	70deg	5
5	Wheel Run-up		0.2-0.5
	Wheel unloading	10rpm	0.2-0.5
	Wheel Stop		5

consists of two sensor heads plus two electronic packages forming two sensor systems completely independent functionally and electrically. Typically, one is active and one is in standby redundancy.

Each sensor has a thin fan-beam-shaped field of view (FOV) of 128°. When tilt-mounted to each other on the spacecraft, two sensor heads cover the FOV of ± 88° with respect to the spacecraft's equatorial plane. Thus the ± 40° portion of FOV is shared by two sensors.

These sensors give "Gray coded" coarse sun angle with accuracy of 1° for full FOV except the central FOV portion of ± 5° where the sensor output is calibrated in 0.1° increment to an accuracy of ± 0.05° by using the digital output and analog sine and cosine outputs. The sun pulse phasing can be obtained with the accuracy of 0.1° for sun angle between ± 40°.

This sun incidence pulse is utilized to time the gas jet thrusting for attitude correction and also to synchronize the picture taking operation of the UV camera.

Star scanner

The star scanner is a passive type with two light sensitive slits placed behind the optical system. One slit is nominally parallel to the spacecraft spin vector and the other inclined 20° with respect to the first. A pair of silicon diode detector elements on a common substrate provides the calibrated star coincidence signals which can be used to determine spacecraft attitude.

A commanded threshold determines the minimum star signal level which will produce output star coincidence signals. An input signal called "spin angle gate command" defines the portion of the spacecraft rotation during which the sensor will respond to stars. This input signal can be used to limit output response to only those spacecraft angles for which stars of interest are expected, thus reducing the probability of unwanted noise events.

The sensor optical axis will be at an angle of 15° to the spin axis of the spacecraft so as to use Canopus (α-carinae) as the main target star for attitude determination.

Momentum wheel

During the picture taking operation of UV camera, DC-motor driven momentum wheel will be used to stabilize the spacecraft motion.

The nominal stored angular momentum is 20 N-m-sec, at 3300 rpm wheel speed. Tacho-pulses picked up magnetically will be used to control and assure the very stable rotational rate of the wheel.

Very quick run-up and run-down operation of the wheel is desirable to shorten the time for picture taking operation during which the communication with the ground may not be maintained.

Gas Jet Subsystem

The unified attitude-control and trajectory-correction hydrazine propulsion system consists of 6 body-fixed 0.3 kg thrusters with Shell 405 catalytic beds that supply trajectory correction, spin axis precession, and spin-up and -down capability. The tank and manifold assembly will be modular with two spherical titanium tanks with bladder inside.

Both N_2 pressurant gas and N_2H_4 fuel are stored in the same tank and blow-down system is adopted.

Fig. 14 shows the system schematically.

The thrusters, two are mounted axially and four are canted on the top side of the spacecraft, apply spacecraft torques as couples, with redundant operational modes as shown in Fig. 15 and 16.

Performance varies with elapsed operational time because of its blow-down system. Table 8 describes the performance at both intial and final conditions.

The range of environmental temperature +5°C ∿ +40°C is required to prevent hydrazine icing and for normal operation of RCS. Tank temperature is controlled by heater system. Thrust chamber is also heated up to 200°C before decomposition of fuel.

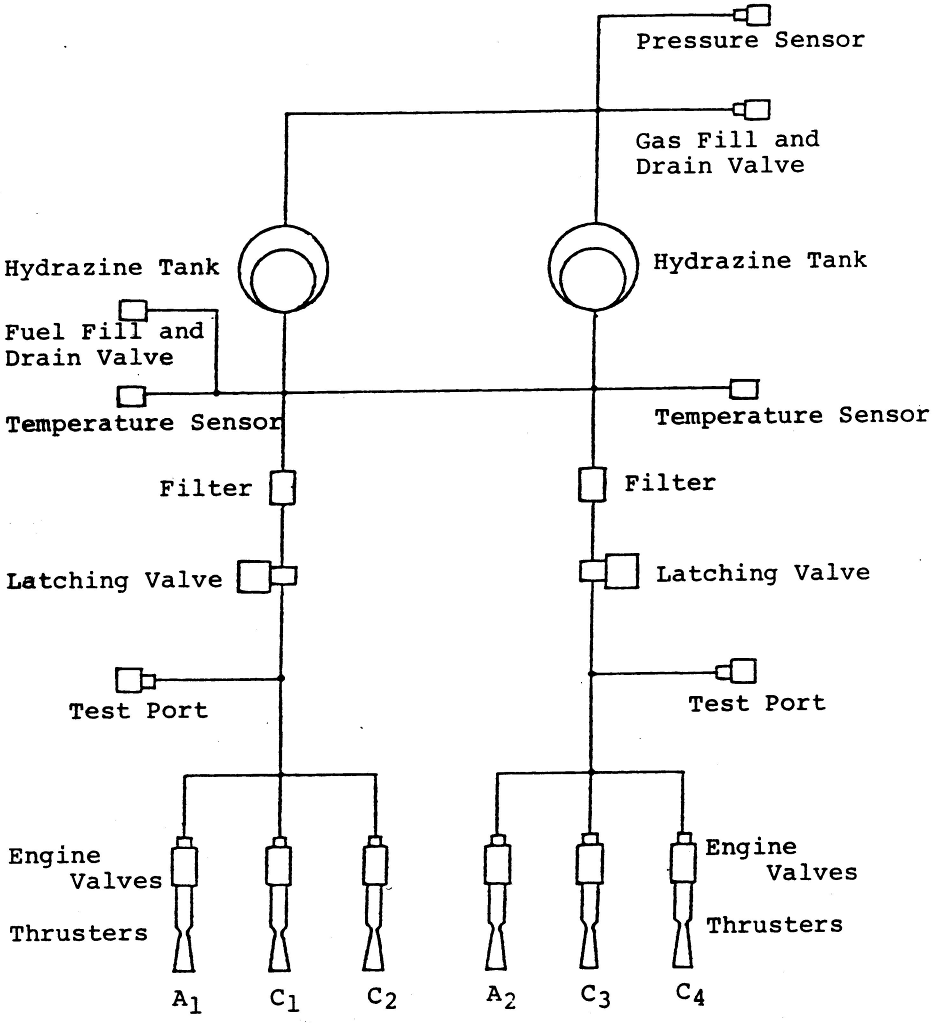


Fig. 14 Block Diagram of RCS

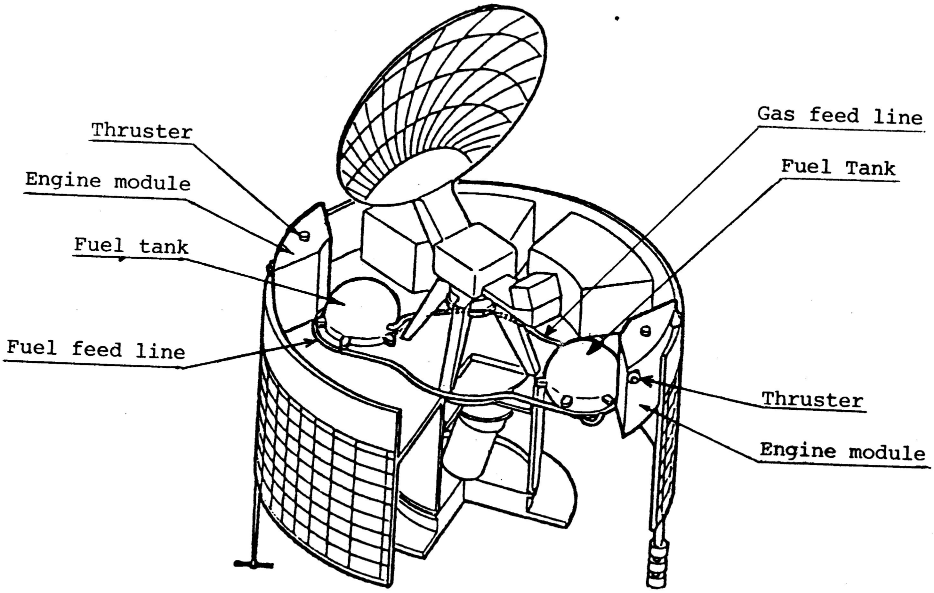


Fig. 15 RCS Configuration

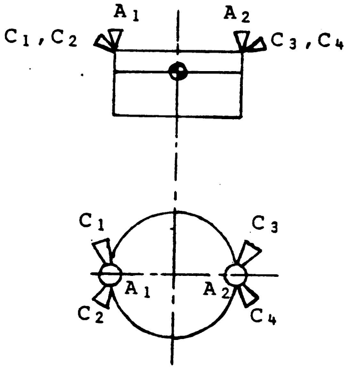


Figure denotes engine no.

Maneuver		Normal	Degraded
ΔV		A_1+A_2	A_1 or A_2
$\Delta\theta$	Coarse	A_1 or A_2	A_1 or A_2
	Fine	$C_1 + C_3$ $C_2 + C_4$	—
ΔN		$C_1 + C_4$	C_1 or C_4
		$C_2 + C_3$	C_2 or C_3

Fig. 16 RCS Engine Arrangement and Operation

Table 8 Performance of RCS

Items	Nomenclature, Units	Initial capability (nominal)	Final available capability	Remarks
1. Thrust	f kg	0.3	0.256	
2. Total Impulse	It kg·s	———	813.	Available total impulse
3. Propellant Weight	W_p kg	5.0	1.0	Available propellant: 80% of total=4 kg
4. Acceleration Force	F kg	0.6	0.512	
5. Precession Torque	T_θ kg·m	0.210	0.179	At , r=0.7 m N=6 rpm $\Delta t=0.414$ s
6. Spin Torque	Ts kg·m	0.347	0.296	Thruster inclination: 60°(Must be verified)
7. Acceleration	$\dot{v}$ cm/s^2	4.36	3.83	Initial total weight: (assumed) Ws/c=135 kg
8. Angular Acceleration of precession	$\ddot{\theta}$ deg/s^2	5.70	4.86	Moment of inertia(assumed): $Iz=3.36$ kg·m·s^2
9. Spin Acceleration	$\dot{N}$ (rpm)/s	0.986	0.841	Ditto
10. Optimum Pulse	Δt s	0.414	0.414	
11. Minimum Pulse	Δtm s	0.100	0.100	Must be verified
12. Optimum Precession Angle per Pulse	$\Delta\theta$ deg	2.46	2.01	
13. Minimum Precession Angle Per Pulse	$\Delta\theta min$ deg	0.57	0.49	
14. RCS Dry weight	W kg	10.0	10.0	
15. Tank Pressure	P_T kg/cm^2	20.0	15.8	

TIMER AND IGNITION SYSTEM

Planet-A ignition system employs a timer system in the spacecraft for its own exclusive use on account of the following reasons.

1) Characteristics of ignition system require specific parts and components.
2) Isolation of system from other events of sequence is favorable for safety.
3) Separate timer-ignition system doesn't bring appreciable weight increase in comparison with employing a common spacecraft timer.

But, if the separation of kick motor could be achieved with other alternative means, it remains possible to mount this timer on the third stage to save the weight resource.

The timer-ignition system activates following items:

1) Ignition of the 3rd stage motor
2) Separation of the 3rd stage motor from the kick stage
3) Ignition of the kick motor
4) Separation of the kick motor from the spacecraft
5) Telemeter channel change

First 4 items concern to ignition system.

The block diagram of system is given in Fig.17.

The principal part of the timer adopts a redundant system with three sets of units consisting of crystal oscillator, counter and sequence memory (ROM) to increase reliability.

Its accuracy is far beyond the required one of ± 0.1 s.

The output is sent out only when two of outputs of three units coincide.

The outputs of the timer energize ignition relays through drivers for the first 4 items. The last one is directly operated by a driver.

The timer starts operation by a signal sent from the rocket-borne timer.

The power source of the timer is provided by stabilizing the bus line DC 24 V, and also the electric power for ignition circuits is fed from a capacitor bank being connected with the same bus line.

The timer stops either antomatically after all the events of sequence are executed or compulsorily with the radio command.

All the outputs of items of sequence are monitored through telemetry.

Fig.18 illustrates interface of the timer-ignition system.

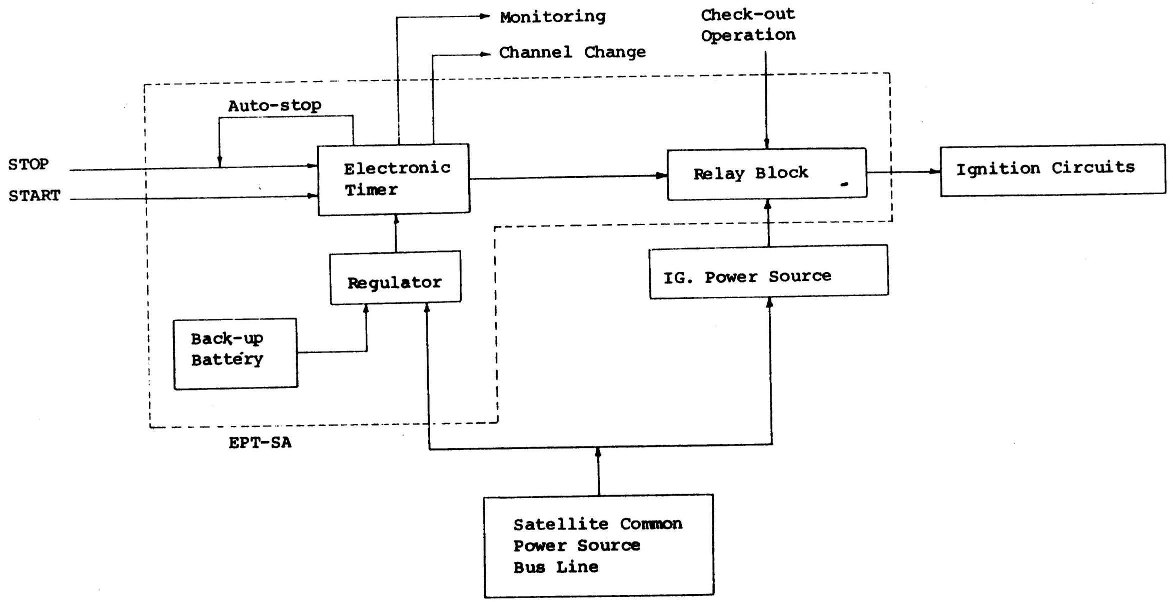


Fig. 17 Timer and Ignition System Block Diagram

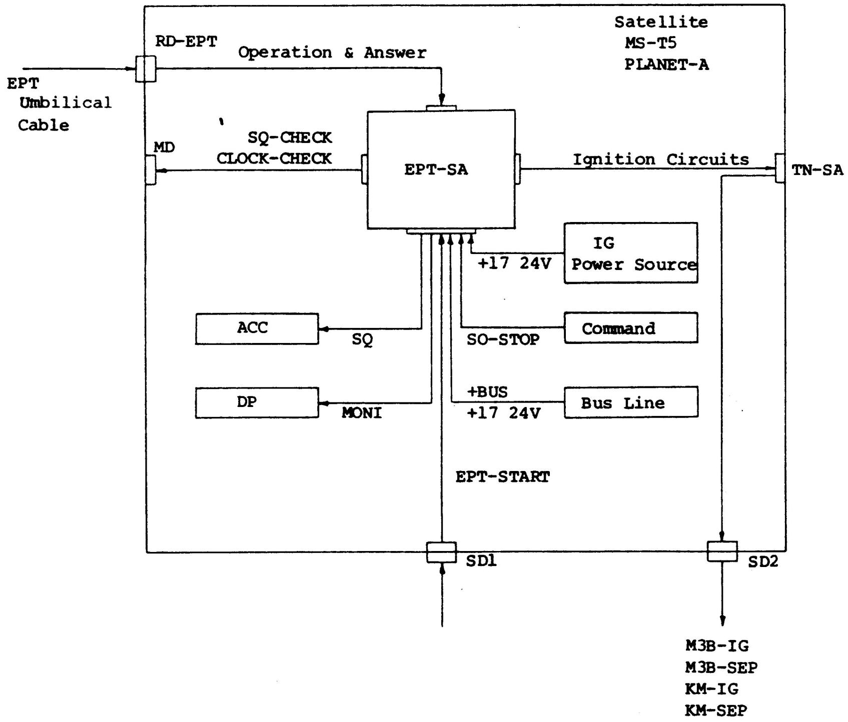


Fig. 18 Interface of the Timer-Ignition System

POWER SUBSYSTEM

The power subsystem onboard Planet-A spacecraft is designed to reliably provide the regulated power to the scientific instruments and common subsystems for a minimum life of one year in the interplanetary circumstances.

The power subsystem consists of a solar panel, a secondary battery, a battery controller, a bus voltage limiter and two DC/DC converters as shown in Fig.19.

The solar panel as the primary source of power, which comprises about 2000 back surface field solar cells with back surface reflector is installed on the exterior surface of the cylindrical spacecraft. Each solar cell size is mainly 2 cm x 6 cm and the size of 2 cm x 2 cm is also used.

The solar panel can supply a power of 67 to 104 watts at the sun distance of 1 to 0.68 AU with above subsystems while spacecraft spin axis is perpendicular to the ecliptic plane.

The secondary battery, which consists of 15 nickel-cadmium cells, delivers the power to the subsystems until the heat shield jettisoned in launch phase, and it is also necessary during the periods of peak power required. Each cell has the nominal capacity of 2 A-h. Commands can be transmitted from the ground station to disconnect the battery from the spacecraft main bus if the battery should malfunction.

The battery controller supplies the trickle charge at the current of C/15 to the battery. And the bus voltage limiter bounds the main bus voltage under 23.5 volts in order to protect bus loads from dangerous high voltages.

The two DC/DC converters receive an input power of 17 to 24 volts and furnish the output power regulated at 5,12, -12 and 28 volts. The frequency of their oscillators is adopted to be 50 kHz for the weight reduction.

The performance of power subsystem is summarized in Table 9.

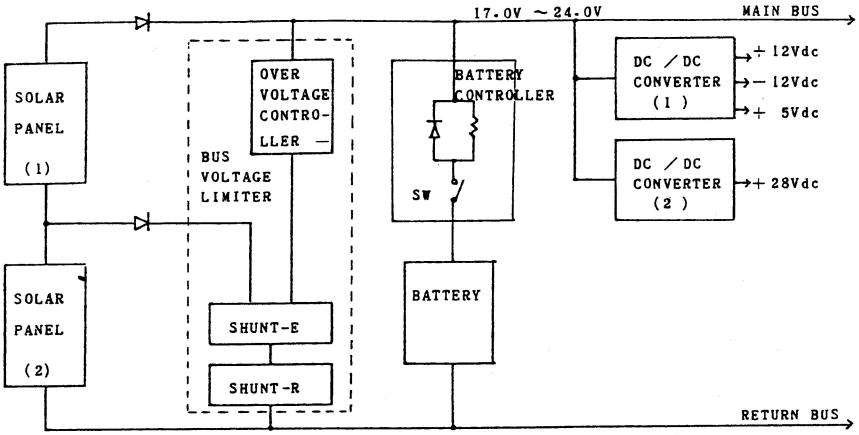


Fig. 19 Power Subsystem Block Diagram

Orbit	Perigee : 0.68AU , Apogee : 1AU
Design life	1 year minimum
Solar panel power	39W ··· $\theta s^{*} = 35^\circ$,1AU 67W ··· $\theta s = 90^\circ$,1AU 82W ··· $\theta s = 90^\circ$,0.83AU 104W ··· $\theta s = 90^\circ$,0.68AU
Power required	36W ··········· $\theta s = 35^\circ$,1AU 46W ·········· $\theta s = 90^\circ$,1AU 70W (80W**) ··· $\theta s = 90^\circ$,0.83AU 43W (78W**) ··· $\theta s = 90^\circ$,0.68AU
Bus voltage	17Vdc ~ 24Vdc
Battery charge control	Charging at the current of C/15
Battery capacity	2A-h
Battery depth of discharge	60 % maximum
Voltage regulation	+ 12Vdc , − 12Vdc , + 5Vdc , + 28Vdc

* The angle between spacecraft spin axis and sunline.
** The value in parenthesis means that during data transmission.

Table 9 Power Subsystem Performance

STRUCTURE

Major structural elements of the Planet-A are a center thrust tube structure with attach rings; an instrument and electronics platform; a solar-array substratum; and a pedestal for the antenna as shown in Fig. 20.

(1) Center thrust tube structure
This is composed of a CFRP cylinder, with the upper and the lower rings which attach the platform and the kick stage rocket engine, respectively; and eight struts supporting the platform.

(2) Instruments and electronics platform
The sandwich construction of aluminum alloy skins with aluminum honeycomb core is adopted.

(3) Solar-array substratum
This consists of the sandwich construction of GFRP skins with aluminum honeycomb core.

(4) Pedestal for the antenna
The pedestal support for the offset parabolic antenna is made of forged aluminum alloy.

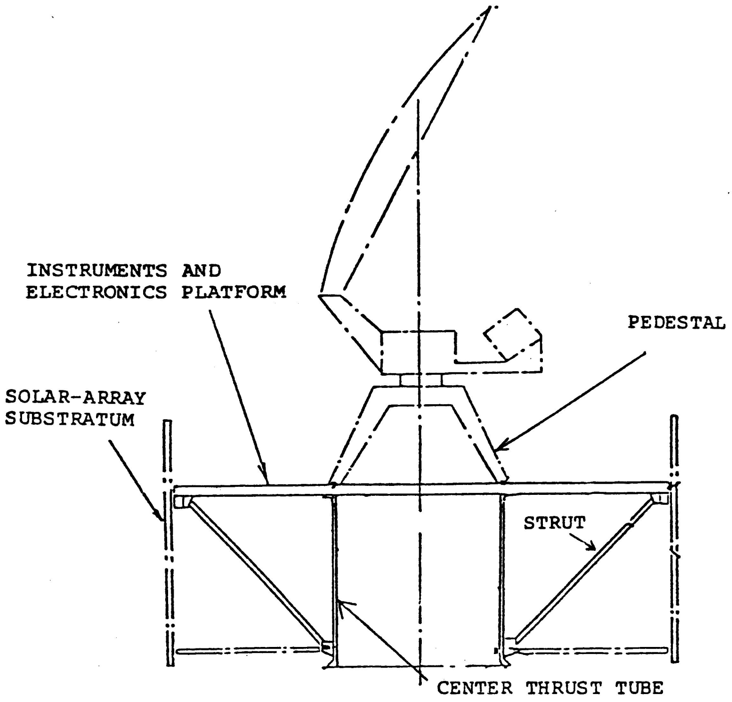


Fig. 20 Structure

THERMAL CONTROL

Design Concept

A general idea of thermal design of Planet-A is to maintain the spacecraft during its entire period of mission life within a reasonable temperature range by means of a suitable thermal subsystem.

The spacecraft thermal environment, which is almost exclusively dominated by the solar radiation energy except for a short period of launch phase, changes significantly along with the distance between the sun and the spacecraft.

The intensity of incident solar energy is constantly increasing from 1 solar in the vicinity of the earth to almost 2.2 solar at the perihelion, and then decreasing to about 1.5 solar at the encounter with the comet.

The principal design concept of Planet-A is to control the heat flow of the spacecraft mainly by means of active control method, or more specifically, by the thermal louver device together with passive method of thermal insulations and coatings as an auxiliary agent.

The design configuration is to cut or minimize the incident solar energy by enveloping the spacecraft outer structure with thermal insulations, letting the excess heat generation inside the spacecraft dissipate to space through the platform surface backed with the louver device.

Design Configuration

Main features of the current thermal subsystem and coatings of Planet-A spacecraft are schematically illustrated in Fig.21.

The louver device is composed of total six fan-shaped segments placed on the back side of platform of which surface is made of second-surface-mirror(SSM) material. Each segment has a capability of regulating heat dissipation within a range of 7.5 watt at the temperature of 10°C to 42.6 watt at 20°C corresponding, respectively, to the complete shut and full open of blades which are sensed and driven by bimetallic actuator.

Accordingly, the effective emissivity of actively-controlled surface which is about 54 percent of total platform area varies from 0.15 to 0.74.

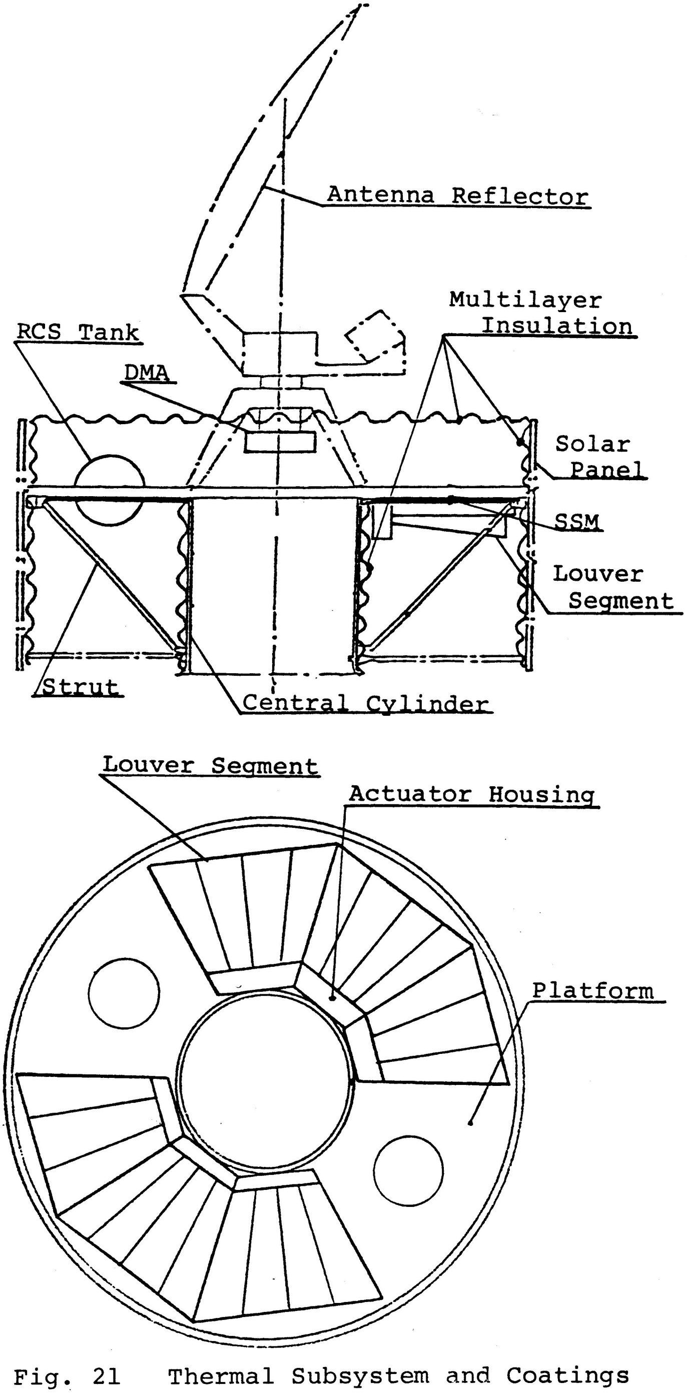


Fig. 21 Thermal Subsystem and Coatings

The thermal insulation is a multilayer blanket of more than 15 sheets of thin aluminum-deposited myler surfaced with comparatively thick kapton sheet.

The RCS tanks and lines are thermally controlled independently from other part of spacecraft by means of electric heater and thermal insulations. The electric heater will be used only at the early phase of cruising before the attitude control maneuver begins when the spacecraft generates minimum power of about 35 watt due to the insufficient sun-spin angle.

A battery is also mounted to augment the solar power not necessarily for RCS heater but also for power consumption in other parts at this early phase.

Most of the important electronic boxes and instrumentations on the platform are black coated to enhance radiative heat exchange among them.

A mechanical despin motor assembly (DMA) of the high-gain antenna is stored inside the spacecraft structure and is thermally decoupled from the reflector which is exposed in space by insulating materials.

ENVIRONMENTAL REQUIREMENTS

MECHANICAL ENVIRONMENT

Mechanical environment of Planet-A during launch phase is evaluated as shown in Table 10 ,11 ,12 , and 13. Shock and vibration should be recognized as those at the interface between Planet-A and its launch vehicle. Since the launch vehicle is under the way of development, these tables may be updated as the desigh advances.

Test

Sinusoidal vibration test, random vibration test and shock test will be performed under the same condition stated above. Each test will be performed separately and furthermore axis by axis. It must be noted that the flight model is to be subjected to those tests before the flight operation.

Table 10 Sinusoidal Vibration

X axis		Y and Z axis	
10∿12 Hz	0.69 mm(0-p)	10∿12 Hz	0.23 mm(0-p)
12∿15 Hz	2.2 mm(0-p)	12∿15 Hz	2.76 mm(0-p)
15∿30 Hz	0.69 mm(0-p)	15∿30 Hz	0.23 mm(0-p)
30∿80 Hz	2.5 mm(0-p)	30∿80 Hz	0.83 mm(0-p)
80∿500 Hz	1.0 mm(0-p)	80∿500 Hz	0.5 mm(0-p)
500∿2000 Hz	5.0 mm(0-p)	500∿2000 Hz	1.67 mm(0-p)

NOTE 1 X axis : thrust axis
Y and Z axis : two orthogonal axes perpendicular to X axis
NOTE 2 sweep rate : 1 oct/min

Table 11 Random Vibration

	X axis		Y and Z axis
5 Hz	0.0063	G^2/Hz	T.B.D.
5∿10 Hz	9	dB/oct	
10∿40 Hz	0.05	G^2/Hz	
40∿81 Hz	-12	dB/oct	
81∿337 Hz	0.003	G^2/Hz	
337∿600 Hz	12	dB/oct	
600∿1500 Hz	0.03	G^2/Hz	
1500∿2000 Hz	-12	dB/oct	
2000 Hz	0.0095	G^2/Hz	

duration : 60 sec.

Table 12 Shock

X axis	+25 G peak half sine	duration : 10 msec
	-10 G peak half sine	duration : 10 msec
Y and Z axis	5 G peak half sine	duration : 10 msec

Table 13 Acoustic level

T.B.D.

MAGNETIC ENVIRONMENT OF MS-T5

As the MS-T5 spacecraft is installed with a sensitive magnetometer to measure the weak interplanetary magnetic field of several tens or less nT, the magnetic contamination of the spacecraft itself at the location of sensor must be reduced to 1 nT and preferably smaller.

Therefore, the material and parts of spacecraft should be non-magnetic as possible as they can.

However, some of them may be inevitably magnetic. So the special attention must be paid to the geometrical arrangement of these pieces to minimize the magnetic effect to the sensor position.

Also electrical circuits is so designed that they do not make magnetic field in the spacecraft.

SCHEDULE OF DEVELOPMENT

The overall development schedule for Japanese first interplanetary flight project is shown in Table 14.

The major feature of this schedule are:

--- Development of proto-model (including structural and electrical model and their assembly, integration and test)

--- Flight models (MS-T5 and Planet-A) development

--- Improvement of Mu launch vechicle and static firing tests of each stage motor

--- Construction of launch pad for Mu-3U

--- Construction of ground tracking station including software.

Schedule of Development

Development of Planet-A Project is as following